F-22 Structural Coupling Lessons Learned

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Abstract

A survey of current F-22 aeroservoelastic analysis and testing activity shows that valuable insight has been gained into several structural coupling and ride quality problems. The aeroservoelastic (ASE) analysis results agree well with flight and ground test measurements. Examples from a recent structural coupling test will be used to illustrate some recent F-22 ASE issues.

Introduction

The nature of the F-22's mission requires a flight control system (FLCS) which is robust at many different flight conditions. The combination of flexible structure, high bandwidth actuators, and high gains in the FLCS guarantees some structural coupling difficulties. Figure 1 shows a picture of the aircraft and its control surfaces. The horizontal tails and thrust vectoring nozzles are used for pitch control. The tails, ailerons, flaperons, and rudders are used in the lateral and directional axes. The FLCS accelerometers are near the cockpit and the rate gyros are about 150 inches aft of the cockpit.

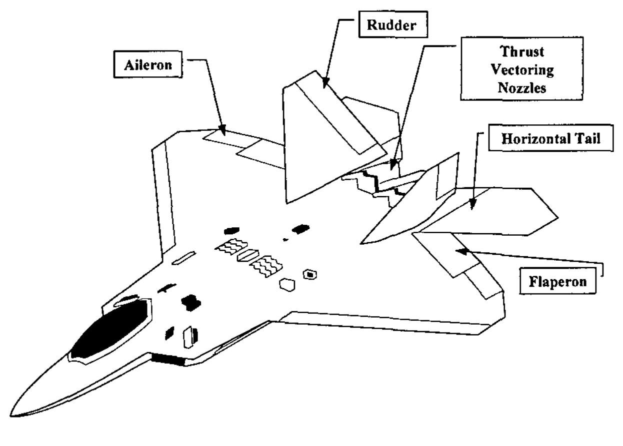

Figure 1 F-22 Aircraft Control Surfaces

Figure 2 shows the F-22 in flight with the Stabilization Recovery Chute (SRC) installed. This is also called the spin chute. It is required for high angle of attack flight testing until adequate spin stability can be shown. A recent structural coupling test was conducted to evaluate the effect of this 1100 pound structure on the critical fuselage bending modes. An additional justification for the test was the loading of a new operational flight program (OFP) with control law changes that had structural coupling ramifications.

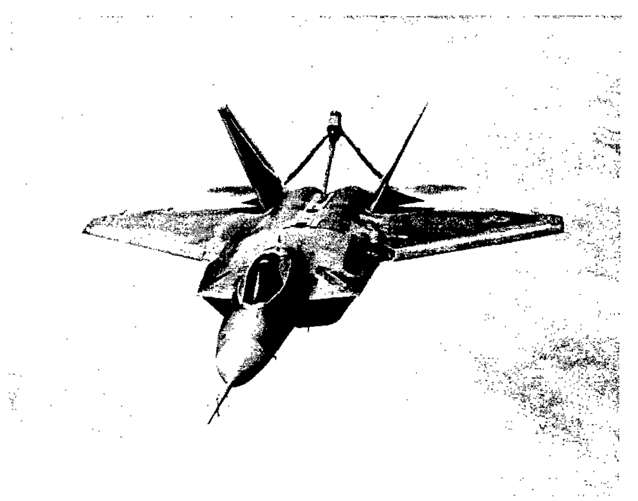

Figure 2 F-22 with Stabilization Recovery Chute (SRC)

Analysis Issues

The lessons learned in the analysis area will be reviewed before proceeding to specific test cases. The analysis issues encountered to date fall into two obvious categories: 1) Modeling the FLCS control laws, and 2) Modeling the structural transfer functions.

The aeroservoelastically sensitive modes on the F-22 are in the 8 to 18 Hz frequency range. There are structural filters on all rate gyro sensor feedback signals and on the vertical and lateral acceleration signals. The goal is to eliminate interaction from the structural modes without causing degradation to the flying qualities due to phase loss and associated time delay.

Modeling the Flight Control Laws

The pitch axis FLCS is fairly easy to model in the analysis. It is essentially a single input, single output system. There are other paths in various parts of the envelope but these have been found to contribute very little to structural coupling. The frequency response bandwidth of the thrust vectoring nozzle is so limited that it can be neglected for the most part. It is necessary to maintain a lookup table for the current pitch

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axis gains as a function of Mach, altitude, and other parameters, but this data is readily available.

The lateral-directional analysis requires a different approach. Figure 3 is a diagram of the multi-input, multi-output lateral-directional FLCS. The lateral-directional FLCS gains change mainly as a function of angle of attack and speed. There are interconnects between the lateral and directional axes to remove roll due to yaw and yaw due to roll. At some angles of attack, surfaces are removed completely from the system. Initial efforts to model this complex system for all flight conditions were not successful.

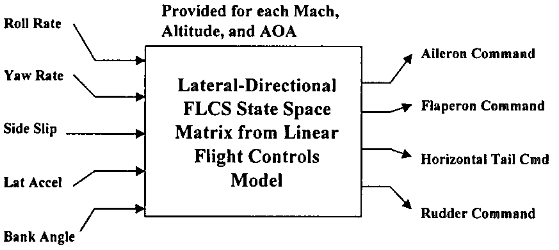

Figure 3 Lateral Directional FLCS for ASE Analysis

Tasking the Flight Controls group with providing state space matrices for each analysis flight condition solved the problem. This requires planning and coordination but it has been quite successful. Figure 4 shows a comparison of the analysis model of the lateral FLCS for a particular path, versus lab test and aircraft ground test for a recent structural coupling test condition. We have found that it is a good 'sanity' check to measure the control law frequency response in the ground simulator and compare this to analysis and aircraft test transfer functions.

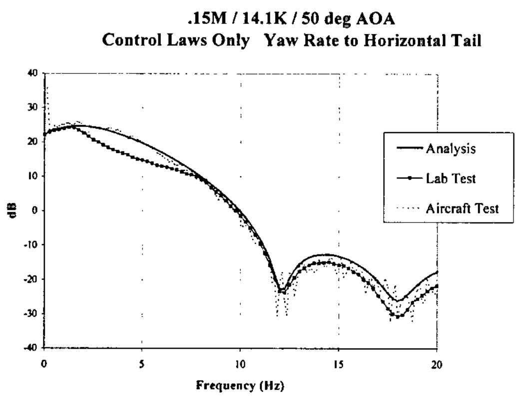

Figure 4 Control Law for Yaw Rate to Tail

Modeling the Structural Transfer Functions

Assuming the control laws are well known, the structure's contribution is the main unknown in structural coupling analysis. The finite element model (FEM) gives estimates of this contribution, but test data is necessary to prove (or disprove) the accuracy of the FEM.

Pitch Axis Structural Transfer Functions

Figure 5 shows the structural transfer function between the horizontal tail and Qb, body axis pitch rate and Nz, the normal acceleration. The strong peak on the charts is the vertical fuselage bending mode. This mode is affected by the overall weight of the configuration. The clean wing condition can have a vertical fuselage bending mode of 10.3 Hz to 11.7 Hz depending on the presence of the spin chute and fuel state.

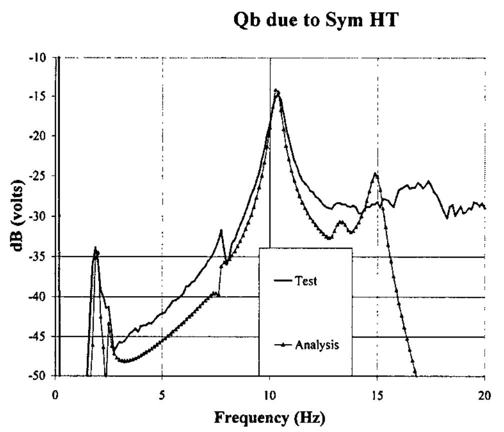

Figure 5a Pitch Rate Structural Transfer Function

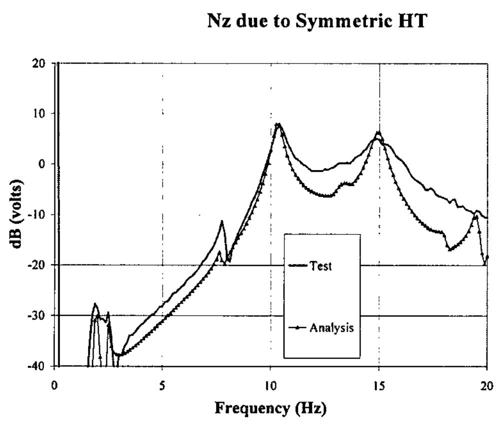

Figure 5b Normal Acceleration Structural Transfer Function

The structural filters for pitch rate and normal acceleration have their maximum effectiveness at 11 Hz. If the fuselage mode is higher or lower by even 1 Hz, the effectiveness is reduced by as much as 10 dB. This will have consequences as the aircraft proceeds through its development program. However, if the structural model has been verified by test, the analyst can make confident predictions about future configuration changes. The analysis matches the test data for the pitch axis cases fairly well. The analysis has been tuned with regard to frequency and damping to achieve this result.

Lateral Axis Structural Transfer Functions

The lateral axis structural transfer functions are shown in Figure 6. Typically, the modes of primary importance for the lateral axis are the wing bending mode at 9 to 10 Hz, and the lateral fuselage bending mode at 14 to 16 Hz.

The analysis predicts the wing bending mode fairly well. The amplitude is close and the frequency is only slightly low. The good analysis correlation allowed the accurate prediction of several roll rate problems that will be discussed in the next section.

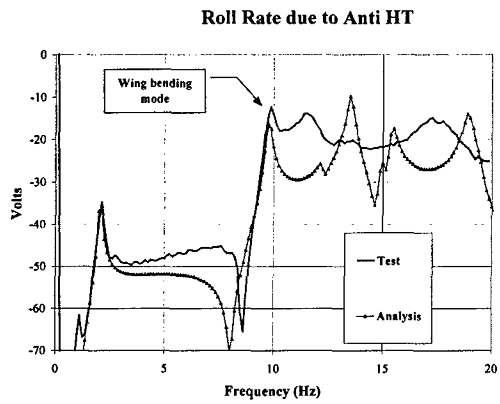

Figure 6a Roll Rate Structural Transfer Function

The analysis is less successful in predicting the amplitude of the lateral fuselage bending mode. This mode is over-predicted by 10 to 20 dB in terms of horizontal tail to Ny, the lateral acceleration at the pilot's seat. This is not the major problem it once was since due to a change in the control laws. The FLCS now uses the lateral acceleration sensor only when the angle of attack is below 16 degrees. When it is important to correctly model the lateral fuselage bending mode for an analysis, the typical practice is to substitute test data in the place of FEM predictions.

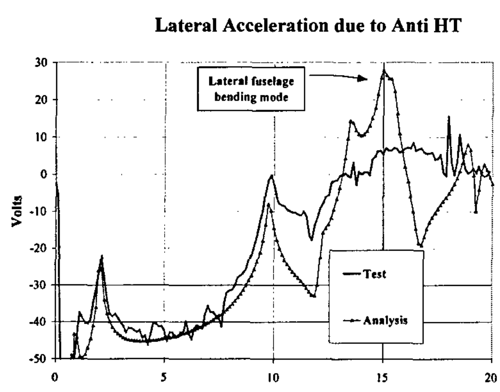

Figure 6b Lateral Acceleration Structural Transfer Function

The lateral and directional axes are controlled using the rudders, flaperons, and ailerons, as well as the horizontal tails. These transfer functions are not shown here but, in general, trends for the other surfaces are similar. The wing bending mode is predicted well and the lateral fuselage bending mode is over-predicted.

Discussion of Specific Case Studies

A recent structural coupling test offers examples of F-22 ASE issues that are of current interest. The main purpose of the test was to provide structural coupling flight safety clearance for the F-22 with stabilization recovery chute (SRC) attached. This 1100 lb installation on the aft part of the aircraft changes the fuselage bending modes slightly. Note that the F-22 structural coupling tests are conducted with the airplane on the landing gear.

The F-22 convention for displaying structural coupling information may not be standard so an explanation is appropriate. Transfer function plots are in dB versus frequency. For test data the Y axis is dB (Volts). The conversion to dB (engineering units) is a constant dB value.

Phase data is not generally reported in the classic Bode style. In general phase considerations have been de-emphasized on the F-22 program. Many feel that the frequencies where problems tend to occur are so high (10 to 20 Hz) that phase predictions are unreliable. The analysis has been tuned to match phase fairly well in the pitch axis, but the lateral-directional still has problems, as will be seen later.

For stability considerations, the first plot shown is generally the open loop transfer function in magnitude form. This plot is dB (dimensionless) on the Y axis, since it is a ratio of output due to an input in the same units. The F-22 has a goal for 6 dB of margin on the open loop transfer function plot, without regard to phase.

When phase is important to show stability, the Nyquist plot is used. The –1 point on the horizontal axis is the neutral stability point and gain margin and phase margin are referenced to this point. The requirement is 6 dB of gain margin and 60 degrees of phase margin.

During a structural coupling test, stability is also shown with closed loop testing. This amounts to simply adding gain to the nominal closed loop system to show required margins.

Discussion of Pitch Axis Test Cases

Pitch Axis Condition #1 "Gravel Road" 160KCAS/1000ft/12deg/Power Approach (Flaps down)

Since the initial flights of the Engineering and Manufacturing Development (EMD) program, the pilots have reported a feeling of light turbulence on approach even in calm air. This has been given the colorful name of Gravel Road, since it feels like the plane is being driven over a rough surface. The dominant frequency is around 12 Hz, the vertical fuselage bending mode. The possibility existed that this rough ride was caused by a structural coupling with the flight control system. Dynamic content was clearly seen in the commands to the actuators but was it cause or effect?

Since the flight test program is currently limited in its ability to measure in-flight stability margins, an experiment was devised to check the level of control system interaction. A switchable filter was created to deepen the pitch rate structural filter on command from the pilot. The idea is that if the control system is causing the Gravel Road, then a deeper filter would improve the ride quality. Figure 7 shows a plot of the flight test aid filter versus the nominal pitch rate filter.

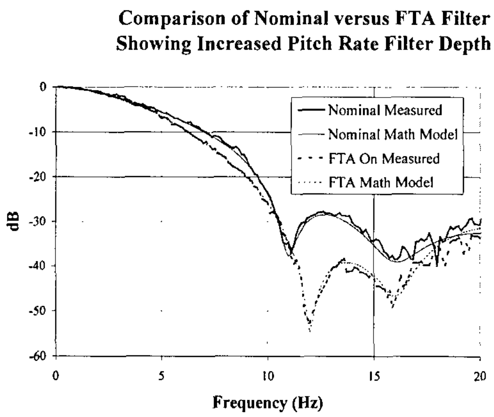

Figure 7 Gravel Road Structural Filter versus Nominal

Figure 8 shows an acceleration time history during the transition from nominal to deeper, flight test aid (FTA) filter. No measurable differences were seen. The prevailing opinion is that the rough ride is caused by separated flow impacting the horizontal tails. However, this issue continues to receive such visibility within the program that every structural coupling test revisits this condition to reiterate that it is not a coupling problem.

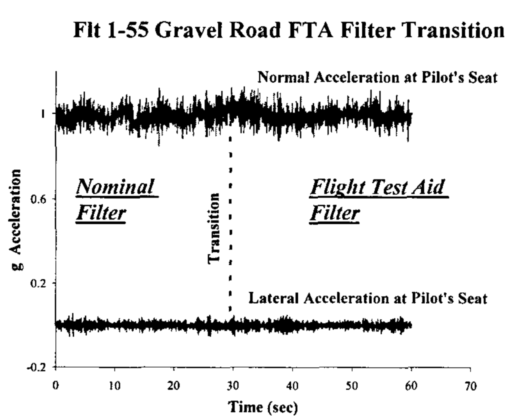

Figure 8 Time History of Pilot Seat Acceleration
Shows no difference with deeper pitch rate filter.

The installation of the 1100 pound spin chute structure causes the vertical fuselage bending mode to go down in frequency by about .7 Hz thus missing the optimal part of the structural filter. Figure 9 shows the magnitude of open loop transfer function with the input at the horizontal tail actuator and the feedback signal to the actuator as the output. The spin chute condition is barely 6 dB down from a magnitude perspective for the Gravel Road condition.

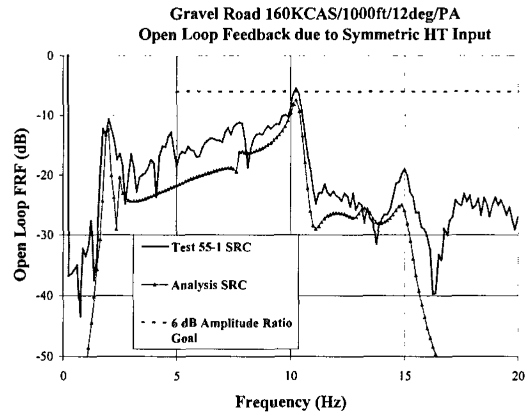

Figure 9 Gravel Road with Spin Chute Open Loop FRF
Shows barely adequate margin

It should be noted that when phase is considered – that is, when a true gain margin is calculated - the actual margin is much greater than the 6 dB shown in the magnitude plot. Figure 10 illustrates this point with a Nyquist plot of the same case.

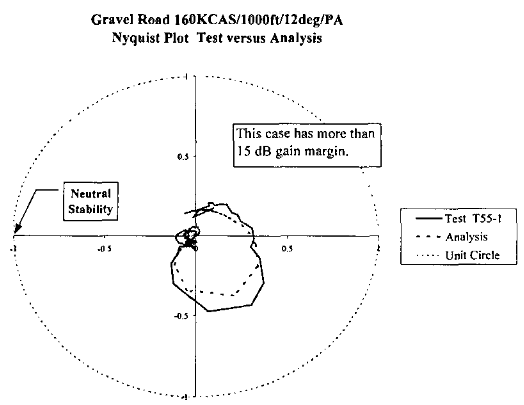

Figure 10 Nyquist Plot for Gravel Road Case
Shows large stability margins.

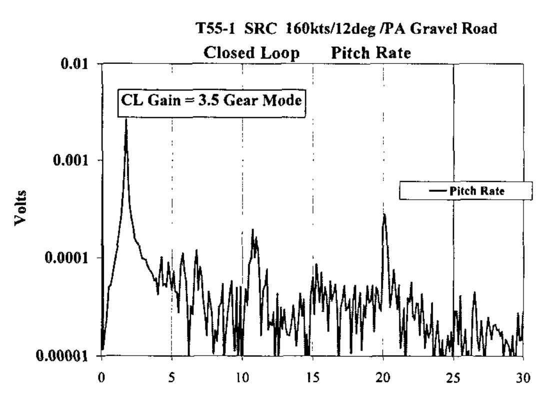

Figure 11 Gravel Road Closed Loop Pitch Rate
With gain of 3.5, only gear mode at 2 Hz is unstable. No problem with fuselage bending mode.

Closed loop testing of this condition also showed a large margin. Figure 11 shows the closed loop pitch rate signal with a gain of 3.5 inserted into the critical path. The next pitch rate case will show the importance of phase considerations.

Pitch Axis Condition #2 .4M/40K/0deg Vector Off

This condition has been tested on every structural coupling test done on the F-22. With vectoring off, all the pitch axis feedback gain is taken by the tail. This results in a 6 dB increase in FLCS gains and a possible decrease in stability margin. Switching vectoring off is a flight test technique only. It will not be possible on production aircraft. It is meant to test the flying qualities where vectoring is inhibited due to an engine anomaly. Also, the vectoring is switched off for parameter identification testing. Analysis predicted that the 6 dB amplitude ratio goal (amplitude margin without regard to phase) would not be met and the test results shown in Figure 12 bear this out.

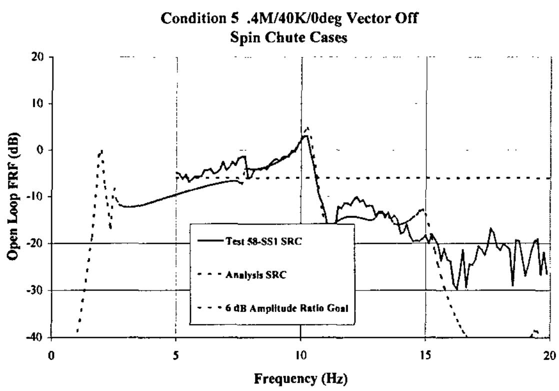

Figure 12 .4M/40K/Vector Off Open Loop FRF

Figure 13 shows the Nyquist plot which accounts for the phase of the open loop frequency response. This plot shows that the high magnitude response is phased such that it will be stable when the loop is closed. This was confirmed with closed loop testing. Despite all testing and analysis showing stability, the vector off case is still approached with caution during flight test. The pilot is briefed that there is potential for a problem and how to respond. So far, the vector off condition has been very stable in flight.

Pitch Axis Condition #3 .95M/High Altitude/0deg Nz Command and Roll Rate Surprise

This condition is called Nz Command because it is the worst case condition for the part of the flight envelope where the Nz sensor is the dominant feedback sensor in the pitch axis. This condition has been tested on all F-22 structural coupling tests.

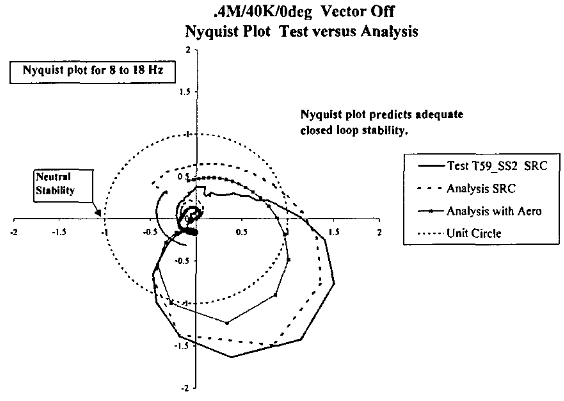

Figure 13 Nyquist Plot of Vector Off Case Showing predicted closed loop stability.

As seen in Figure 14, the spin chute case has a large peak which does not meet the 6 dB magnitude margin goal. This is due to the previously discussed issue of missing the 'sweet spot' of the Nz structural filter because of a lower fuselage bending mode frequency.

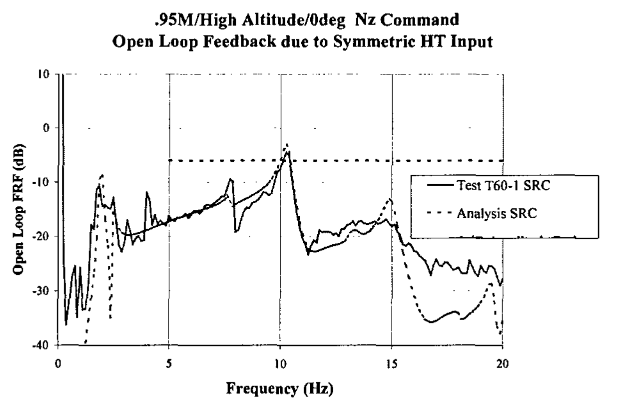

Figure 14 Magnitude Plot for Nz Command Case

The Nyquist plot and the closed loop testing show this mode to be stable. See Figure 15.

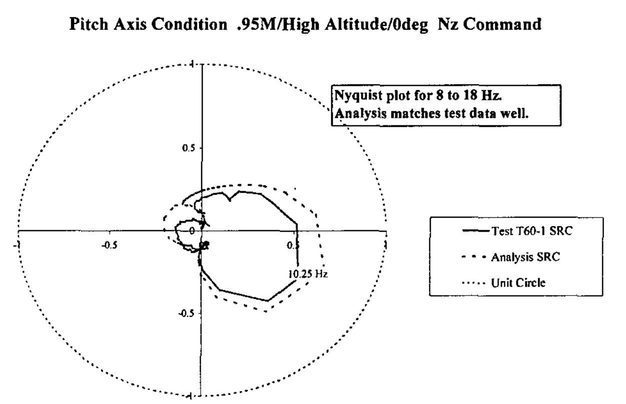

Figure 15 Nyquist Plot for Nz Command

During closed loop testing, an unexpected antisymmetric roll rate instability at 10 Hz was apparent when the gain to the tails was increased. The data was recorded and subjected to post-test analysis. Indeed, the roll rate to horizontal tail loop was only marginally stable at this condition. Figure 16 is the Nyquist plot for roll rate constructed in the post-test analysis.

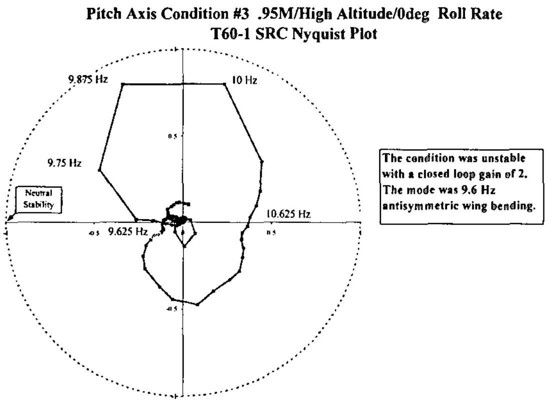

Figure 16 Nyquist plot for Roll Rate at Pitch Condition #3

Figure 17 is the closed loop roll rate signal measured after a gain of 2 was inserted into the horizontal tail actuator path. The large peak is antisymmetric wing bending. The coupling mechanism is horizontal tail exciting roll of the fuselage, which excites the antisymmetric wing bending mode, which generates roll rate feedback, which generates more horizontal tail motion.

After the problem was understood, the next question was: "How did this condition slip by the ASE Analysis certification process?". Discussions with the flight controls engineers revealed that there is a local peak in the roll rate to tail feedback gains at the .95M/High Altitude condition. The .90M/High Altitude condition had been analyzed and found stable as part of the ASE certification process, but the .95M gains were 9 to 12 dB higher. In addition, the gain is increased by the 0 deg angle of attack of the test condition with respect to a trim alpha condition. The problem is

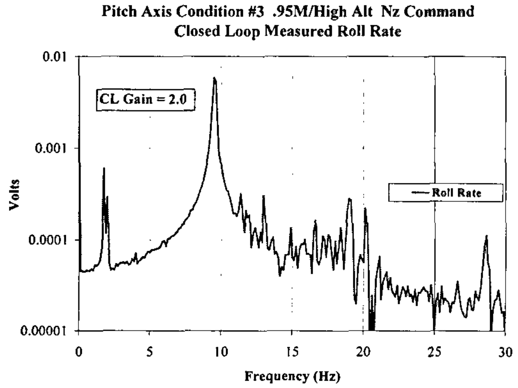

Figure 17 Closed Loop Roll Rate at Pitch Condition #3
Shows unexpected roll rate response.

exacerbated by the fact that in this low alpha flight regime, a shallow 2[nd] order roll rate filter is employed. In retrospect, this is a very good test condition for roll rate.

The flight controls engineers agreed to reduce the roll rate gains at .95M/High Alt to arrive at values which yield 6 dB of stability margin. This was accomplished with a software change request. In addition, the .95M condition will be added to the ASE Analysis Certification plan.

Pitch Axis Condition #4 Pilot in the Loop

During early flights of the first two development aircraft a "pilot in the loop" structural coupling was observed. This was seen during turns when the pilot was applying aft stick while being subjected to a load factor of about 2 g's. One pilot reported that he could feel himself coupling with the aircraft's structural mode. Figure 18 shows flight test data illustrating this coupling. The frequency is about 13 Hz, slightly higher than the vertical fuselage bending frequency.

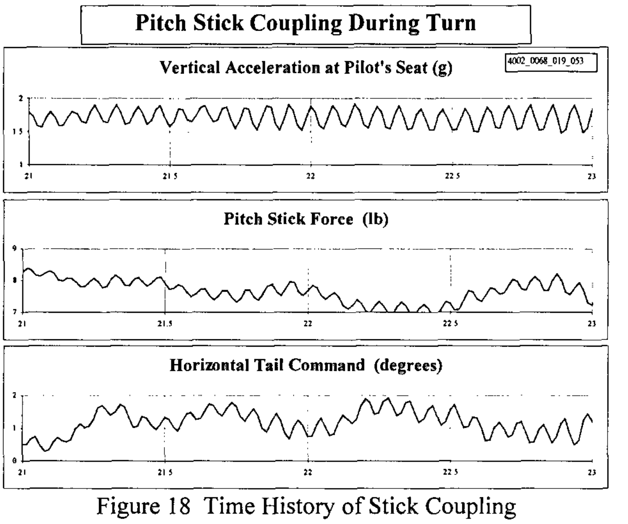

Figure 18 Time History of Stick Coupling

A simple analysis model was constructed to understand the problem. A pilot "gain" was estimated by computing the transfer function between the acceleration at the pilot's seat and the resulting stick force. For the case above, the pilot exerted about .3 lb for every g of acceleration at the 13 Hz frequency. The analysis model confirmed that a problem existed for a portion of the flight envelope.

A structural filter was designed for the pitch stick path that created adequate margin for all test and analysis cases. The filter design had to be coordinated closely with the flight control engineers since the response of the stick is very important to the way an airplane feels to the pilot. Certain overall system time delay requirements dictated a filter with very little phase loss. Throughput requirements set a limit on the filter order. A simple notch filter design met all requirements.

The pitch stick filter was tested to demonstrate its adequacy during the structural coupling test. A volunteer was placed in the cockpit and instructed to apply about 5 lb of aft stick. Data for the unfiltered condition existed from a previous test.

Figure 19a shows the open loop transfer function for the filtered design versus the unfiltered for a worst case condition. The filtered design still has a large response at the fuselage bending mode.

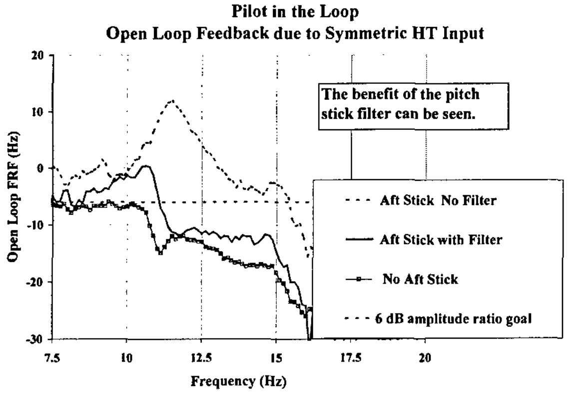

Figure 19a Pilot in the Loop Transfer Function

Figure 19b shows the Nyquist plot for the filtered versus unfiltered test data. This solves the mystery as to why the frequency of the instability was higher than the fuselage bending frequency. The phase of the response caused the instability to be shifted away from the peak response frequency. The plot also demonstrated that the filtered design has more than 90 degrees of phase margin. No "pilot in the loop" problems have been reported in flight testing since the installation of the filter.

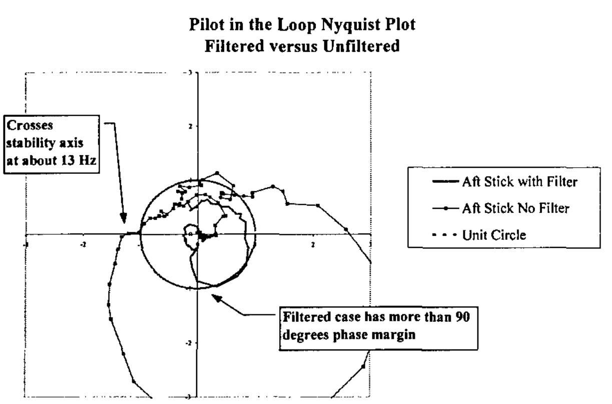

Figure 19b Nyquist Plot for Pilot in the Loop Case

Lateral Directional Test Cases

Lateral Directional Condition #1 .3M/30K/26deg 26 Alpha Concern

Previous structural coupling testing with an older set of flight control laws showed a potential problem at this flight condition due to the contribution of the Ny (lateral acceleration) sensor. When the Ny sensor was opened during testing, the problem went away entirely. The gains at this condition are a strong function of angle of attack. The new operational flight program (OFP) eliminates the Ny sensor for feedback when the angle of attack is greater that 16 degrees. Analysis shows the new FLCS to be very stable at this condition. The test was designed to demonstrate this stability so aircraft limitations could be lifted.

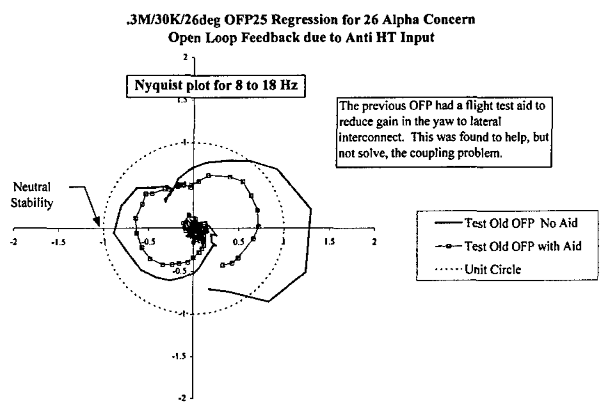

Figure 20a Nyquist Plot for Previous Control Laws Shows beneficial effect of flight test aid.

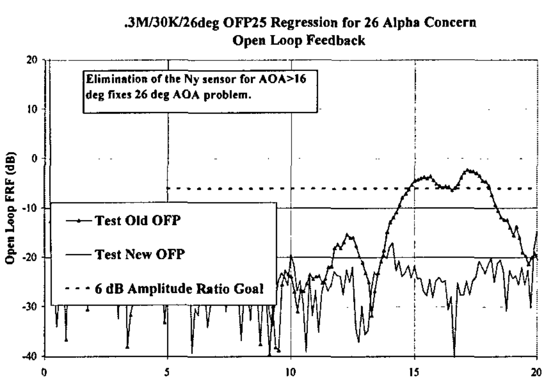

Figure 20b Stability Improvement due to OFP Change

Figure 20a shows a Nyquist plot of previous results. A flight test aid was used to improve the stability as a temporary solution. As expected, the new OFP results are a great improvement over the previous results. See Figure 20b for a magnitude comparison of new versus old control laws at this condition. The 26 degree AOA condition is very stable now.

Flight testing at the 26 degree AOA condition has been marked by a rough ride which has led to pilot comments. At first some thought the entire problem was due to the ASE sensitivity with the old control laws. Certainly, the Ny sensor was feeding significant dynamic content to the actuators. At the time of testing, it was not possible to do an in-flight evaluation of ASE stability margins. Figure 21 shows that the rough ride at the 26 alpha condition has not improved with the new control laws. Apparently, there is considerable buffet at this condition that is not related to the control system.

This rough ride seems to be confined to the 26 degrees and 20000 feet altitude region. If the aircraft goes higher in angle of attack or altitude, the buffet at the pilot's station subsides. At lower altitudes, the buffet does not increase. The bufet has been linked to a vertical tail mode which is excited by separated flow coming from the nose of the aircraft. This vertical tail mode is at about 17 Hz which is very close to the lateral fuselage bending mode. Figure 22 shows the difference in the amount of dynamic content being fed to the actuators due to this buffet-induced signal.

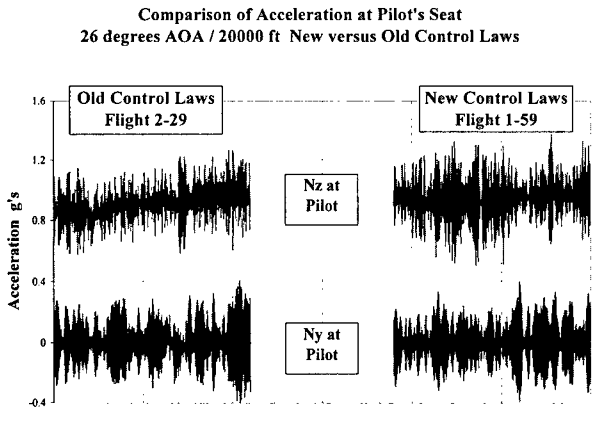

Figure 21 Pilot Seat Acceleration at 26 degrees AOA
New versus Old Control Laws

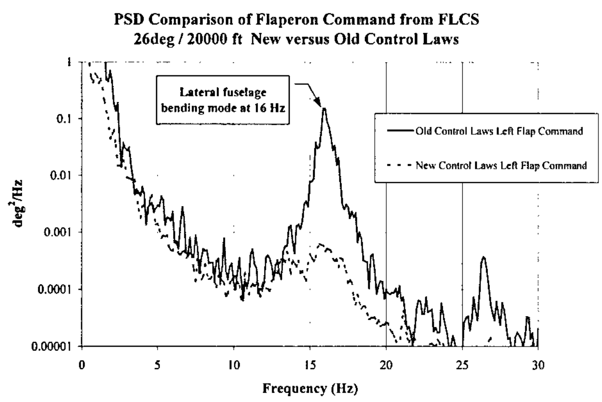

Figure 22 PSD Comparison of Flaperon Command
New versus Old Control Laws

Figure 23 shows the hydraulic pressure in two of the control surface actuators at the 26 degree condition. The time slices shown are the same as in Figure 21. Whether the previous FLCS had an ASE problem or not is still debated, but there is no doubt that the pressure fluctuations seen by the actuators have been reduced dramatically due to the elimination of the lateral acceleration sensor. This is bound to be beneficial to their service life.

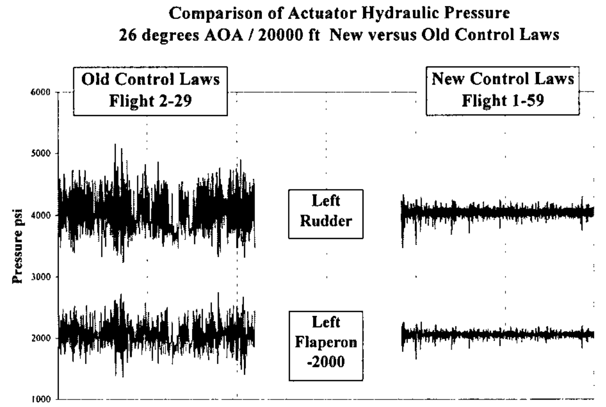

Figure 23 Comparison of Actuator Hydraulic Pressures
New versus Old Control Laws for 26 degree AOA

Lateral Directional Condition #2
.25M/26K/60deg Max Roll Rate to HT

This condition was chosen by a survey to determine the worst case roll rate gains for the flight controls update. Pre-test analysis showed the case to be marginally stable. The critical mode is 10 Hz wing bending. The horizontal tail gets the airplane rolling, which excites the antisymmetric wing bending, which imparts roll rate to the roll rate sensor, which commands more horizontal tail.

Figure 24 shows an example of the control laws correlation for this case. Extreme high AOA cases are more difficult to simulate on the F-22 because the flight data is being received from the inertial reference system. This requires good test technique on the part of the control system hardware engineers.

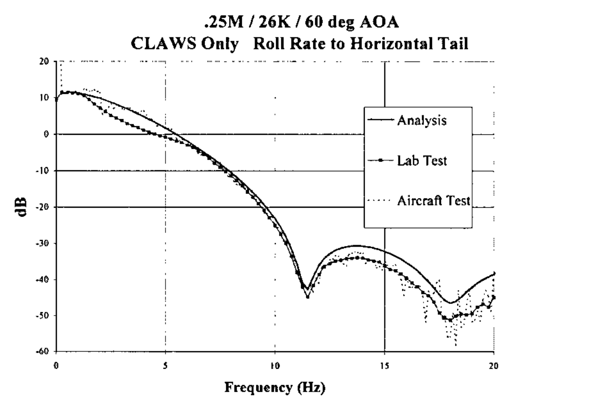

Figure 24 Controls Laws Correlation for Max Roll Rate Case

The test results for the spin recovery chute (SRC) case, shown in Figure 25, agree well with pre-test predictions with regard to the critical 10 Hz wing bending mode. Figure 25 is an example of sensor input test data. The loop was opened at the roll rate sensor and the transfer function is ratio of the output to the random input.

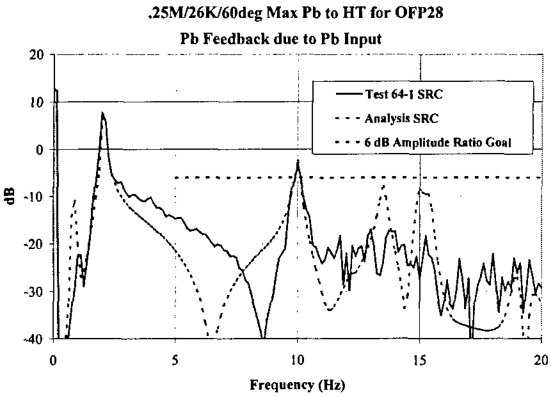

Figure 25 Max Roll Rate Case Magnitude Plot

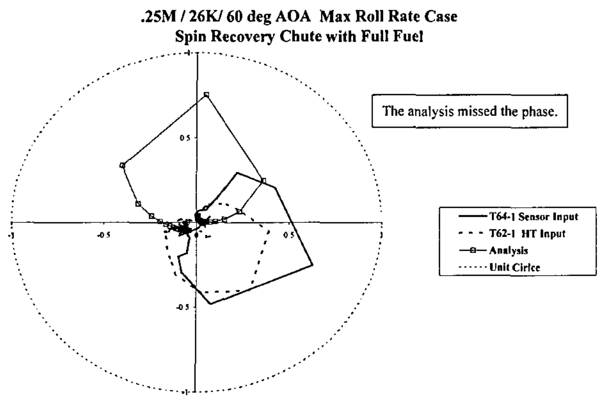

Figure 26 Nyquist Plot for Max Roll Rate Case
Comparison of Sensor versus Surface Input and Analysis.

The Nyquist plot in Figure 26 also shows that the analysis missed the phase. The analysis will be tuned to predict the phase more accurately for cases of this type.

It is helpful to simplify a multi-input, multi-output system to a single-input, single-output system. Pre-test analysis indicated that this flight condition was dominated by the horizontal tail to roll rate sensor path.

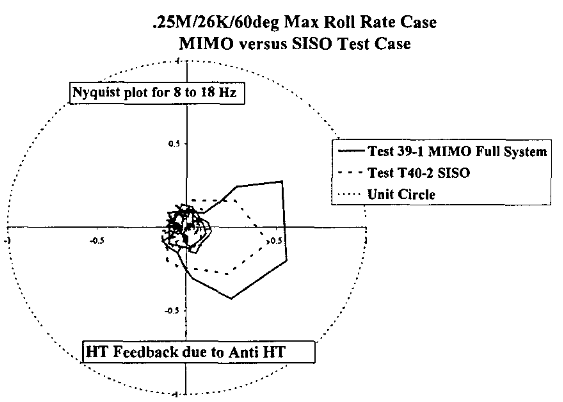

Figure 27 Nyquist plot for Max Roll Rate Condition
Full System versus Single Input, Single Output

To show this, the test case was run in a single input, single output (SISO) condition. In this test, the roll rate sensor is the only active sensor and the horizontal tail is the only active surface. The results are within 1.5 dB of the fully functional FLCS result. The Nyquist plot for the SISO system versus the full system is shown in Figure 27.

The closed loop testing for this condition was not possible in the conventional sense. The horizontal tails were being nearly saturated using artificial pitch rate to keep the FLCS on condition. This left no margin for applying more gain in the horizontal tail path. Though it was not possible to increase the gain, the nominal closed loop case was shown to be stable as expected.

Summary

The F-22 ASE methodology has evolved to a level of maturity that is adequate to show safe flight. The ground and flight testing confirms and agrees with the analytical models. The foundation of results obtained to date will help solve new problems as the aircraft continues through its development program.

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